

TRAJECTORY DESIGN ANALYSIS OVER THE LUNAR NODAL CYCLE FOR THE MULTI-PURPOSE CREW VEHICLE (MPCV) EXPLORATION MISSION 2 (EM-2)

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The first crewed mission, Exploration Mission 2 (EM-2), for the MPCV Orion spacecraft is scheduled for August 2021, and its current mission is to orbit the Moon in a highly elliptical lunar orbit for 3 days. A 21-year scan was performed to identify feasible missions that satisfy the propulsive capabilities of the Interim Cryogenic Propulsion Stage (ICPS) and MPCV Service Module (SM). The mission is divided into 4 phases: (1) a lunar free return trajectory, (2) a hybrid maneuver, during the translunar coast, to lower the approach perilune altitude to 100 km, (3) lunar orbit insertion into a 100 x 10,000 km orbit, and (4) lunar orbit loiter and Earth return to a splashdown off the coast of Southern California. Trajectory data was collected for all feasible missions and converted to information that influence different subsystems including propulsion, power, thermal, communications, and mission operations. The complete 21-year scan data shows seasonal effects that are due to the Earth-Moon geometry and the initial Earth parking orbit. The data and information is also useful to identify mission opportunities around the current planned launch date for EM-2.

INTRODUCTION

Exploration Mission 2 (EM-2) ushers in humans return to exploring beyond low Earth orbit. The Multi-Purpose Crew Vehicle (MPCV) Orion will be the spacecraft used to transport humans to a lunar orbit and back to the Earth safely. Orion will be launched into space on the Space Launch System (SLS) and sent to the Moon using the Interim Cryogenic Propulsion Stage (ICPS).

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MISSION TRAJECTORY DESIGN

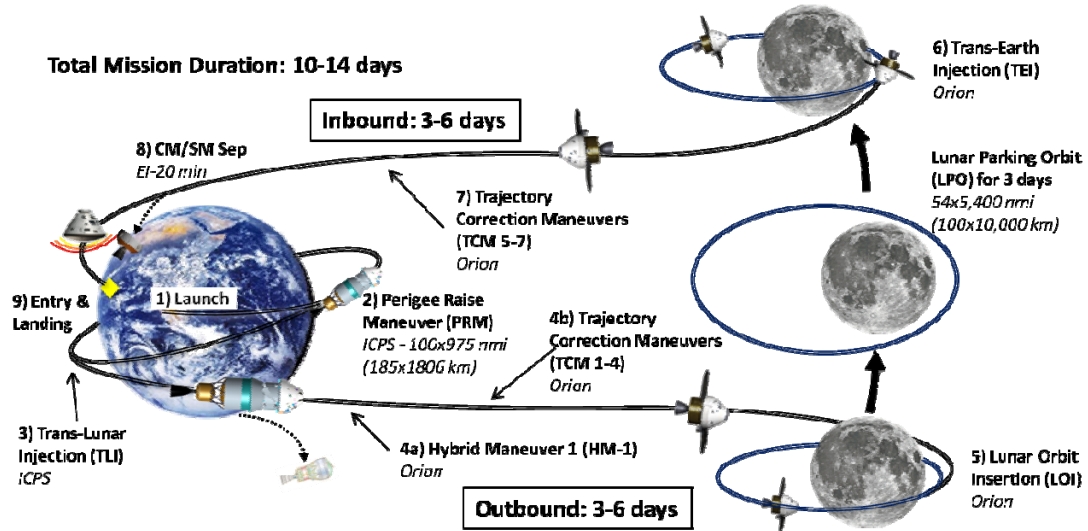


Figure 1. EM-2 Mission Overview

Refer to Figure 1 for a mission overview. The ICPS-MPCV stack is placed into an elliptical Earth orbit of 22x1805 km altitude. When the stack reaches apogee, the ICPS performs a Perigee Raise Maneuver (PRM) to increase the altitude to 185 km. On the second orbit, the ICPS performs the Trans-Lunar Injection (TLI) that places Orion on a free return trajectory back to the nominal Entry Interface (EI) condition. The reason for the free return is that if the Orion Service Module (SM) engine is not able to ignite, then the crew can return to Earth without performing any major maneuvers. The ICPS separates from Orion soon after TLI completion. At a minimum of 4 hours after TLI completion, the Orion SM performs Hybrid Maneuver-1 (HM-1) to lower the lunar flyby altitude to 100 km. When Orion reaches perilune, the SM performs a Lunar Orbit Insertion (LOI) into a 100 x 10,000 km altitude orbit. If LOI is not performed, then after a minimum of 1 hour after the missed LOI, Hybrid Maneuver-2 (HM-2) is performed to target the Earth return. If LOI is successful, then the Orion crew loiters in the lunar orbit for a minimum of 3 days. After 3 days, the SM performs a Trans-Earth Injection (TEI) for the Earth return.

Earth Return

The trajectory targets an Earth EI that places Orion on a splashdown off the coast of Southern California. 6th order polynomials for azimuth and longitude are used to curve-fit through the cluster of EI points. The equations are functions of latitude. The altitude and flight path angle are constants, while the entry velocity is variable.

METHODOLOGY

To understand variations in EM-2 mission design a large 21-year scan was performed by using MATLAB scripts and the Copernicus Trajectory Optimization Program.

Since a lunar nodal cycle is 18.6 years long, it was decided that seven 3-year scans could be performed in a reasonable amount of time and sufficiently cover the lunar nodal cycle. This allowed the seven parts of the overall scan to be processed on independent nodes of the computer cluster simultaneously. The scan dates cover a period from 1/1/2015 to 1/1/2036. The earlier year of 2015 was chosen so one can observe data trends starting with the Moon at minimum inclination and leading up to a maximum inclination in 2025.

The process used to perform the 21-year scan for the EM-2 High Lunar Orbit (HLO) mission involved breaking up the overall mission into logical mission phases. The mission phases are solved sequentially and independently from one another to generate a sub-optimal, feasible mission. The first mission phase is the Lunar Free Return to an Earth entry target line. A solution was computed for each day in the 21-year time period. To reduce the total time of producing the complete 21-year data set, only the epochs that the ICPS has enough performance to complete the mission were used to analyze the remaining mission phases. The second mission phase is the post-TLI hybrid maneuver that lowers the flyby perilune altitude to 100 km, and the missed LOI hybrid maneuver to return to an Earth entry target line. The third mission phase is the lunar orbit insertion into a 100x10,000 km altitude orbit. The final mission phase is the 3 day loiter and TEI to an Earth entry target line. This was a complicated mission phase to evaluate and required computing an impulsive DV solution first, then converting it to a finite burn solution.

After the initial scan database was completed, additional MATLAB scripts were created to take the Copernicus input files and perform more data-mining or post-processing tasks as requests were made for data that were not part of the original dataset. For example, Solar Alpha (also known as the true anomaly of the Sun's projection on an orbital plane) was requested, but is not computed in Copernicus. Figure 2 shows how the Solar Alpha angle is computed during post-processing. Thus, every time a new output parameter is desired, there is no need to re-run the original trajectory database creation process. In addition, Excel and Visual Basic for Applications (VBA) macros were created to perform data collection, time-history file post-processing, and producing chart plots.

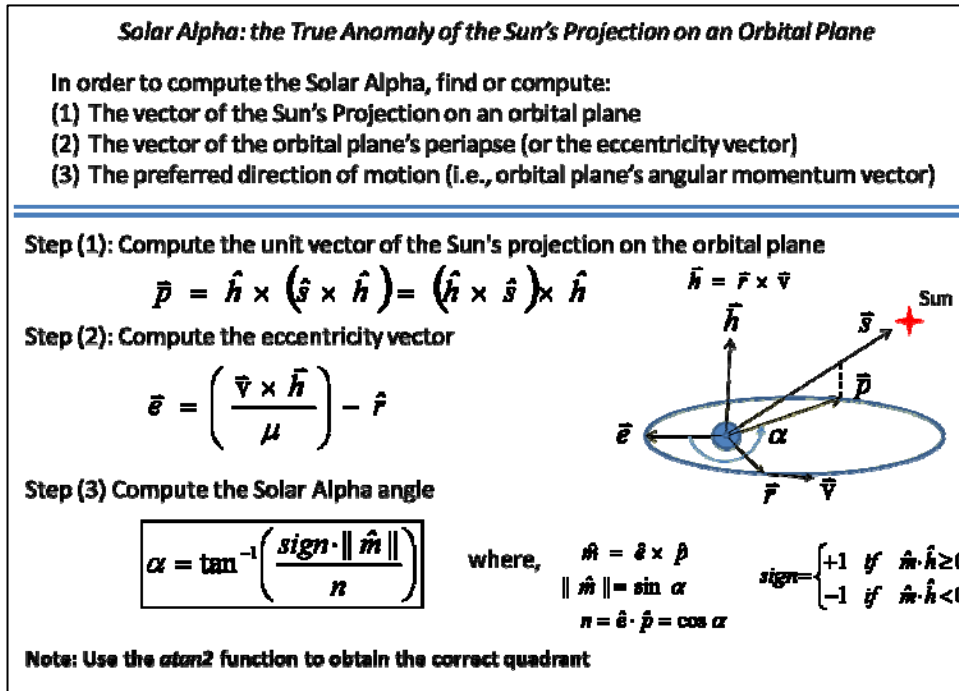


Figure 2. Solar Alpha Calculation

Finally, the initial scan database is only a first pass and consists of sub-optimal roundtrip lunar missions. Further optimization can be made to improve MPCV performance and possibly ICPS performance. One such way is to reorient the lunar parking orbit to optimize the combination of the HM-1, LOI, and TEI maneuvers.

RESULTS

Various vehicle performance and orbital flight parameters were either extracted or post-processed from the 21-year dataset. These data include, but are not limited to, finite burn DV, propellant mass, declination, solar alpha and beta, local launch and landing times, eclipse durations, and state vectors at major events. The data shown are only for valid EM-2 mission ~~opportunities~~ opportunities, which mean the missions satisfy the ICPS and Orion performance capabilities.

The total ICPS DV performance is limited to 2900 m/s for the PRM and TLI maneuvers. The total Orion DV performance is limited to about 1262 m/s for the HM-1, LOI, and TEI or HM-1 and HM-2 (when LOI is not performed). Figures 3 through 7 show the DV performance. Since the Orion-ICPS is inserted into an elliptical Earth parking orbit (EPO), the orbital perigee is in the northern hemisphere for a launch from KSC. Thus, the ICPS mission performance is satisfied only when the Moon's declination (at lunar arrival) is in the Earth's southern hemisphere.

The elliptical EPO limits mission opportunities to about 13 or 14 consecutive launch days and at least 13 or 14 consecutive no-launch days. The consecutive launch days ~~gets-are~~ further reduced by Orion's capability to perform its maneuvers. Since the EPO is elliptical and the perigee is in the northern hemisphere, seasonal effects may occur for other parameters, such as Earth launch or landing lighting.

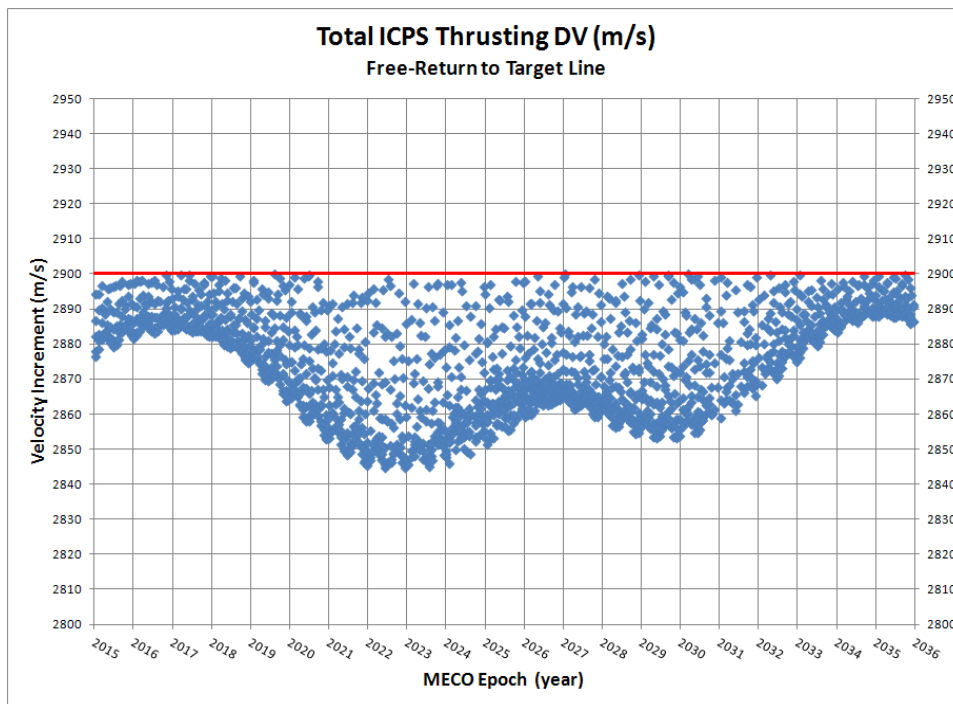


Figure 3. Total ICPS Thrusting DV

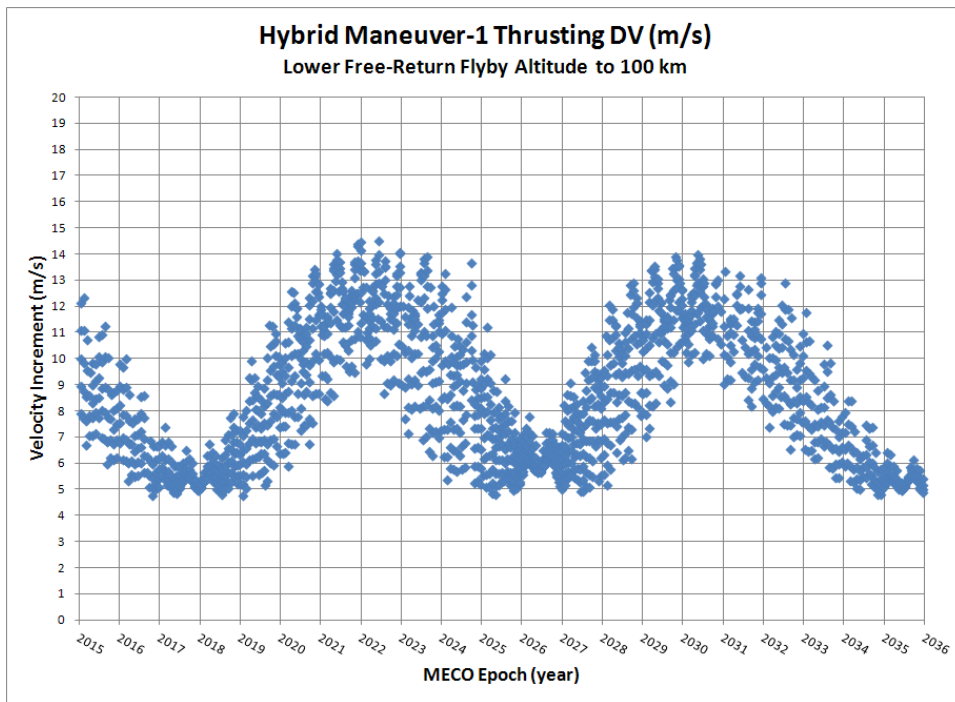


Figure 4. Orion Hybrid Maneuver-1 Thrusting DV

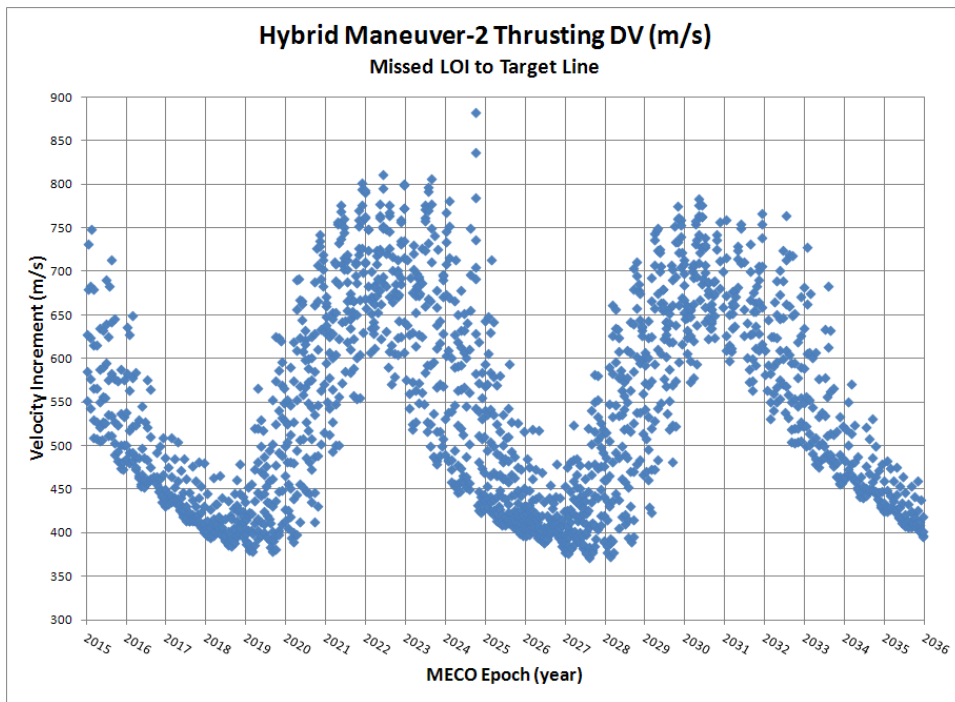


Figure 5. Orion Hybrid Maneuver-2 Thrusting DV

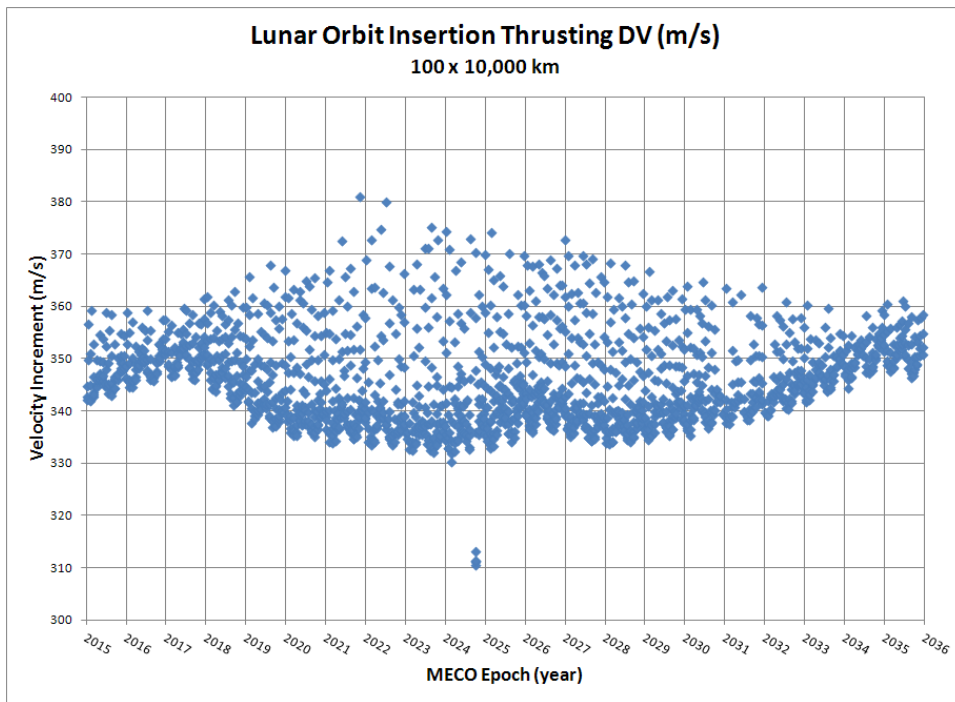


Figure 6. Orion LOI Thrusting DV

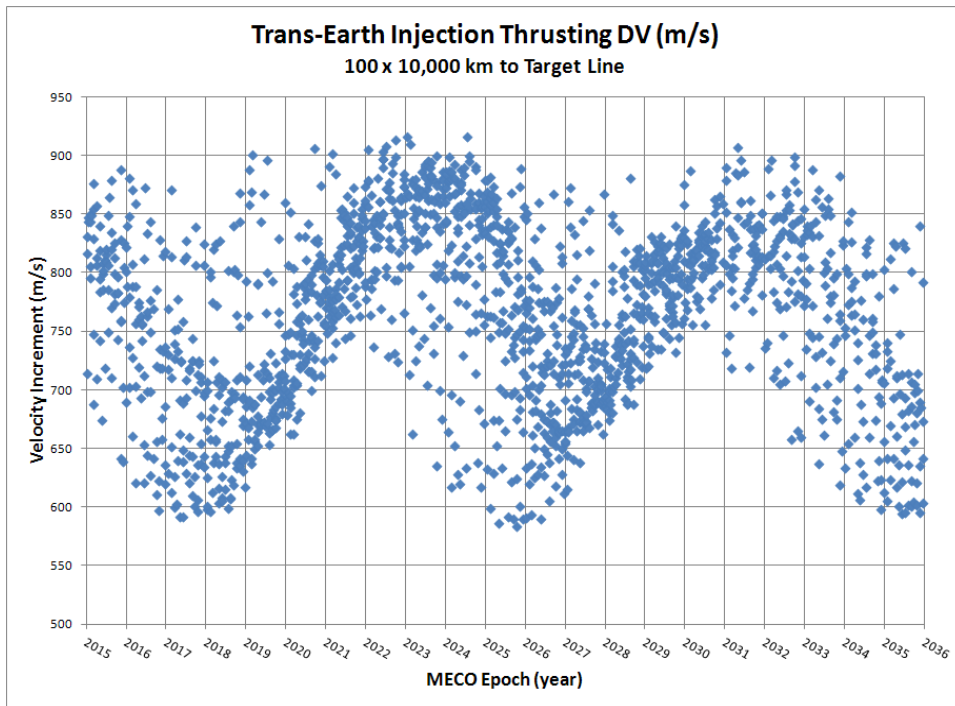


Figure 7. Orion TEI Thrusting DV

The Sun-Spacecraft-Earth (SCE) angle is required by the communication system. The backend of Orion is always pointed towards the Sun for power generation. The communication phased array antenna points away from the backend and has a certain cone angle (e.g., 20, 30, 40, or 60 degrees). Communication with Earth is only available when the SCE angle is within the cone angle. A seasonal effect is present for the SCE angle. Figure 8 shows that the SCE angle reaches its minimum during the summer and maximum during the winter. The angles are approximated by the Moon's orbit inclination relative to the ecliptic plane, which is about 5 degrees. The calculation is shown in Figure 9. Figure 10 shows the SCE angle data at LOI and TEI times for the year 2021.

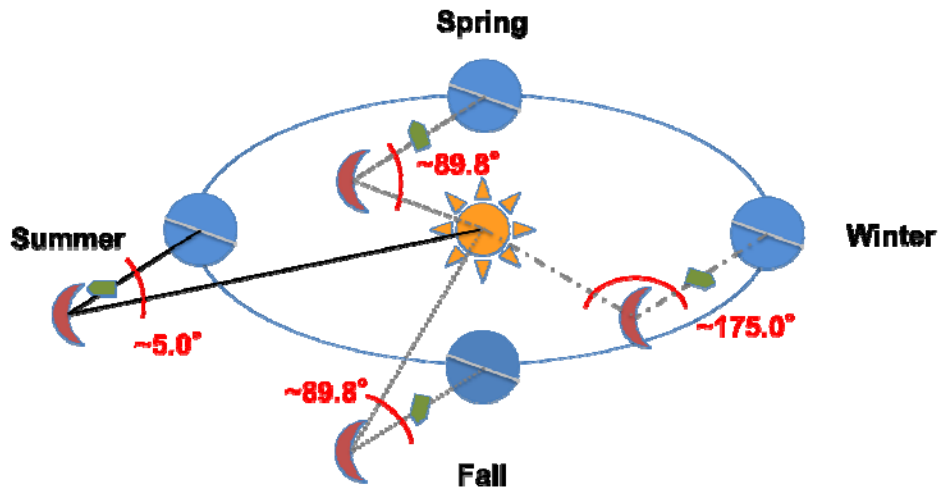


Figure 8. Geometry of Sun-Spacecraft-Earth Angle (seasonal)

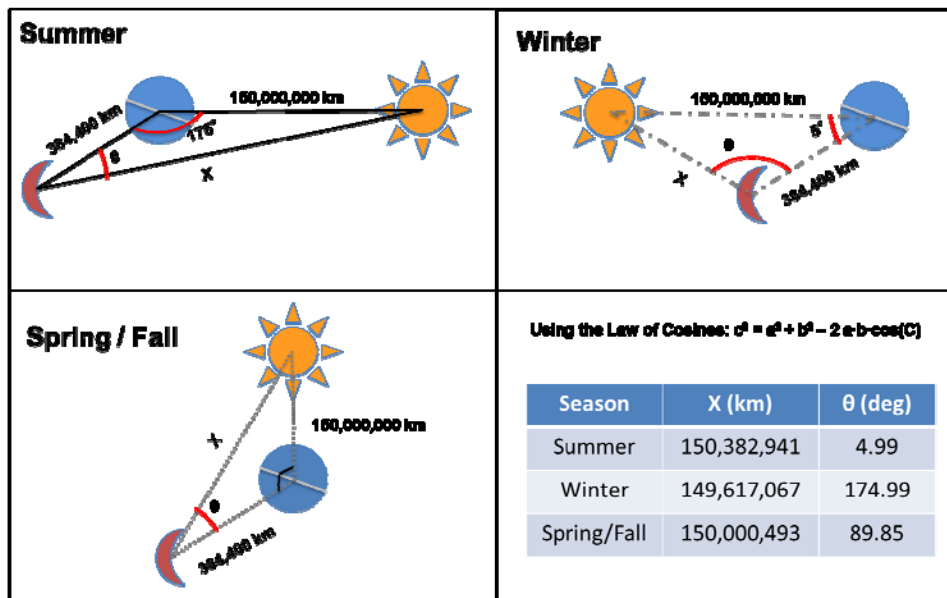


Figure 9. Sun-Spacecraft-Earth Angle Approximation (seasonal)

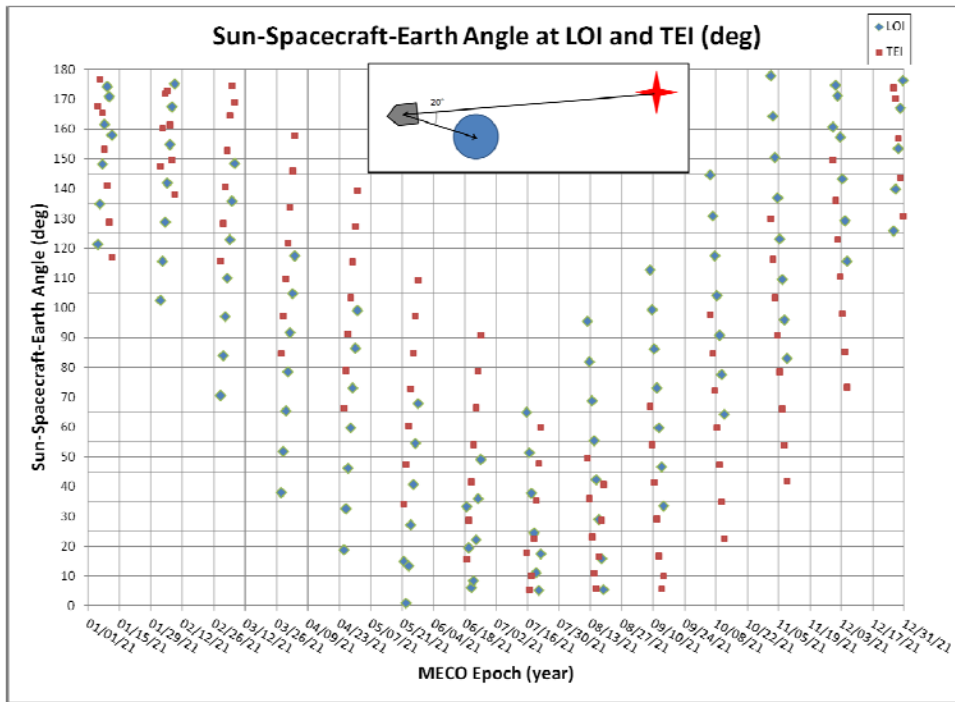


Figure 10. Sun-Spacecraft-Earth Angle for 2021

For mission operations, an understanding of lighting at launch and landing is required. A seasonal effect for lighting is present. During spring to fall months (peaking in the summer), launch and landings can occur in daylight. During the fall to winter months (peaking in the winter), launch and landings can occur at night. The spring and fall months are transitional months can have combinations of night and day for launch and landings. Figure 11 shows the Sun-Earth-Moon geometry for lighting. Figure 12 shows local times for launch and landing along with their respective sunrise and sunset line.

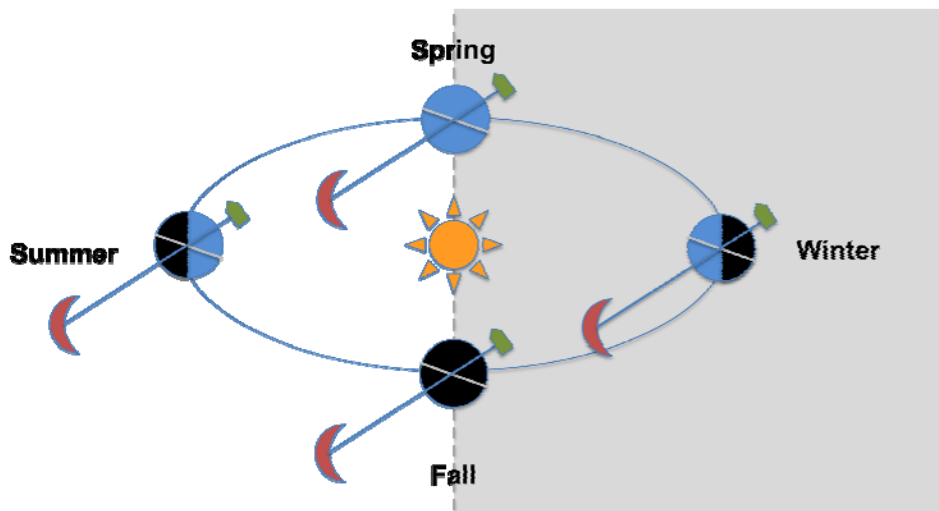


Figure 11. Geometry for Earth Launch/Landing Lighting (seasonal)

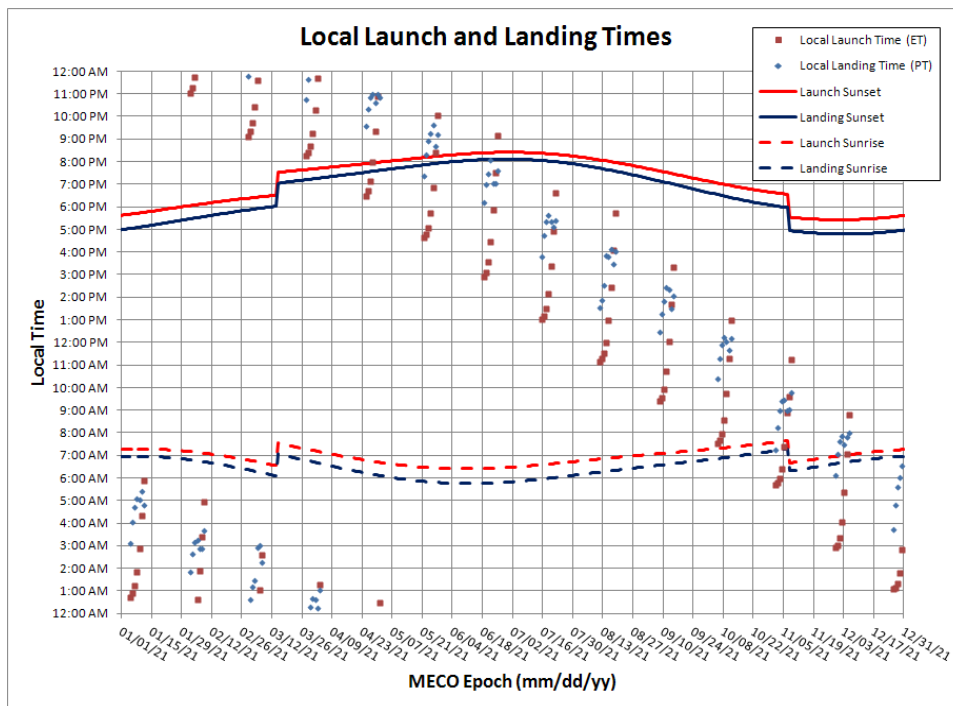


Figure 12. Earth Launch/Landing Times for 2021 (seasonal)

For the power system, information about eclipsing (shadowing) and sunlit periods are required. Figure 13 shows information about how long the last shadowed event was and how long Orion has been in light just prior to performing TEI for the year 2021. The shadow period can be as long as 3 hours, with sunlit periods as long as 13 hours. However, there are times (in August to December) when TEI is performed in the Moon's shadow.

Statistically, probability curves can be associated with the length of eclipse periods. Figure 14 shows a cumulative probability curve for the last eclipse duration at 20 minutes prior to EI. Over the entire ~~21-year~~ 21-year database, about 50% of the missions had no eclipse during the inbound transit. About 94% of all missions, the eclipse duration is 1 hour or less.

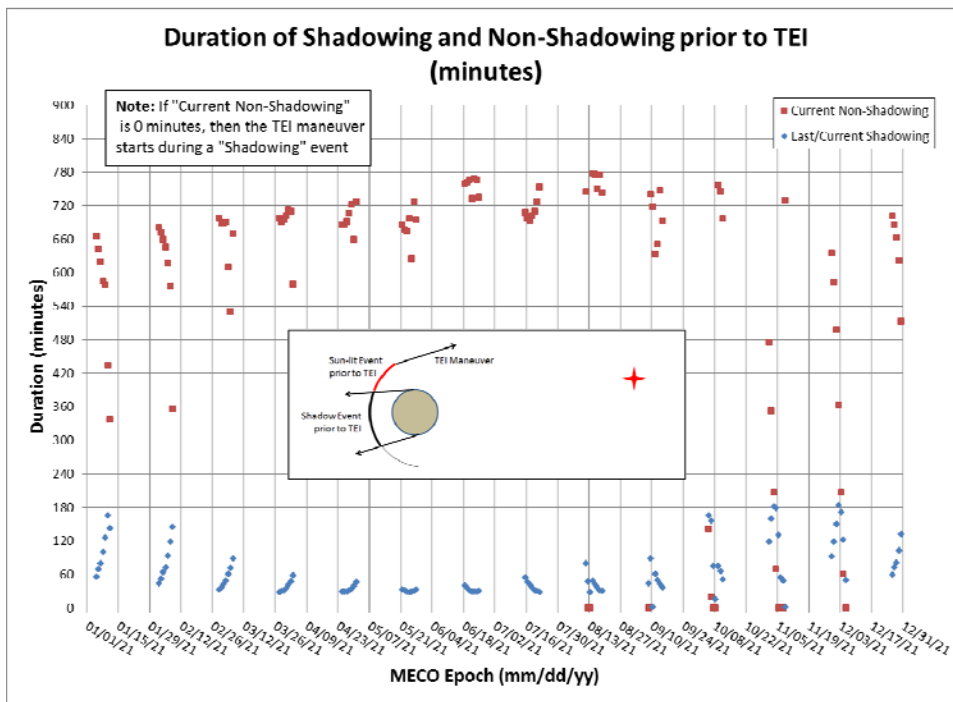


Figure 13. Shadow and Non-Shadow Duration prior to TEI for 2021

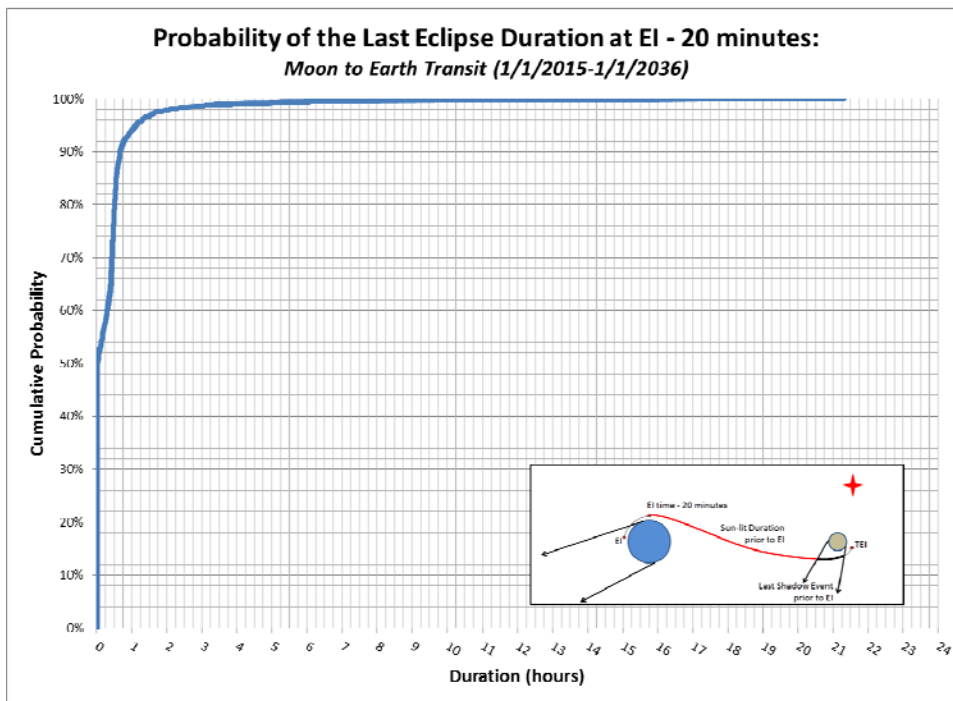


Figure 14. Cumulative Probability of Last Eclipse Duration at 20 minute prior to EI

Other information can also be extracted or post-processed from the database. For example, Figure 15 shows the length of time the Moon is in the Deep Space Network (DSN) Field-of-View

(FOV) when Orion is between the Earth and Moon. When the Moon is in the DSN FOV, communication is not available with Orion.

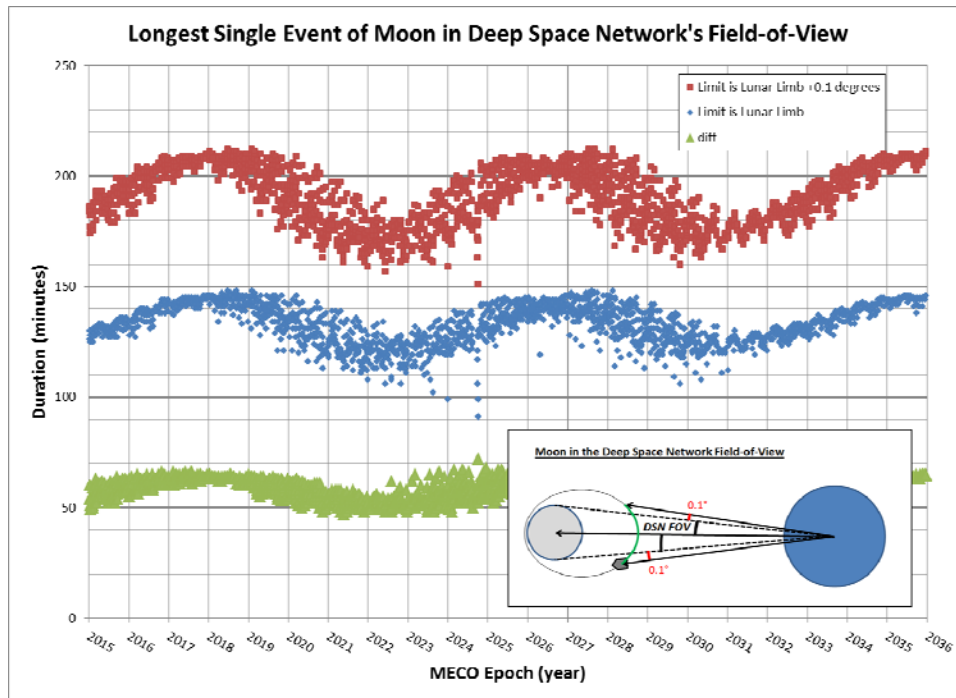


Figure 15. Moon in DSN field-of-view

APPLICATION

The generic 21-year scan results were used by various Orion subsystem teams to determine how their current designs were stressed. Examples of the impact to the following subsystems will be addressed: Communications, Power, and Thermal.

Communications

The exo-Low-Earth-Orbit (LEO) communications support for Orion (i.e. beyond Tracking Data Relay Satellite System (TDRSS) availability) is provided by the Deep Space Network (DSN) sites for communications and possibly some supplemental sites for tracking but not communications. The DSN sites are located in California (Goldstone), Spain (Madrid), and Australia (Canberra). The latitudes of these sites (Goldstone and Madrid in the northern hemisphere and Canberra in the southern hemisphere) lead to the combined coverage capability shown in Figure 16. The figure shows a region over the south Atlantic ocean and southern South America continuing with a sliver up through the Indian Ocean and Asia, the size of which gets smaller as the altitude increases toward lunar distances (although doesn't completely disappear). The shape of this communications blackout zone is created by only having a single southern hemisphere DSN site.

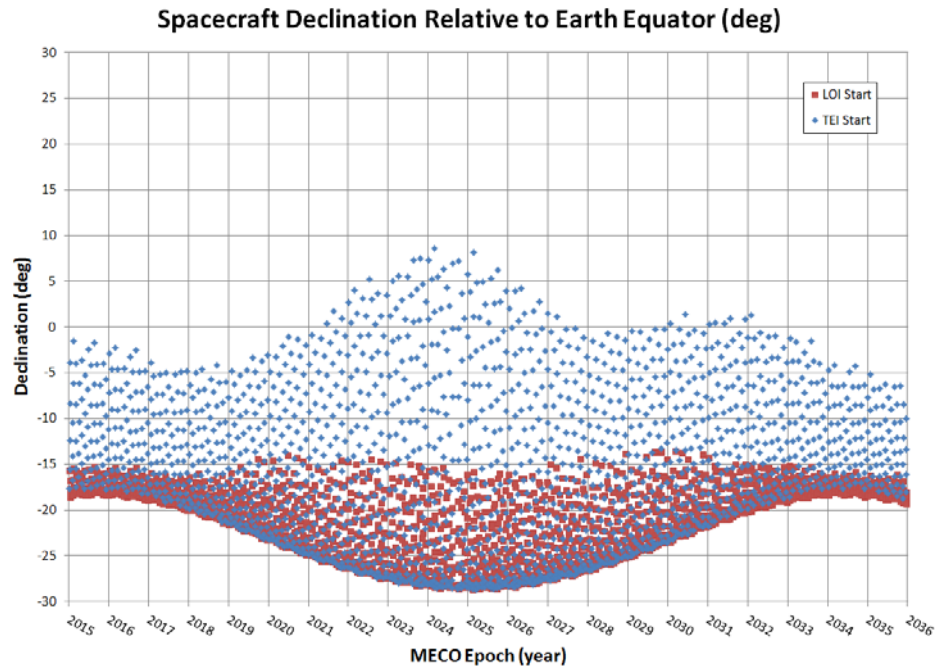


Figure 17. EM-2 Spacecraft Declination w.r.t. to Earth Equator

Lunar Blockage of Earth Line-of-Sight

In addition to communications being limited by DSN coverage communications can also be physically blocked by the Moon. Figure 18218 shows the longest single lunar blockage of Earth line-of-sight for every EM-2 mission opportunity, which range from 33 minutes up to 43 minutes. These blockages are dependent on the lunar orbit properties. For example, EM-2 consists of a repeating high lunar orbit and each orbit will likely contain a blockage very close to the mission max shown in the figure with very little variation.

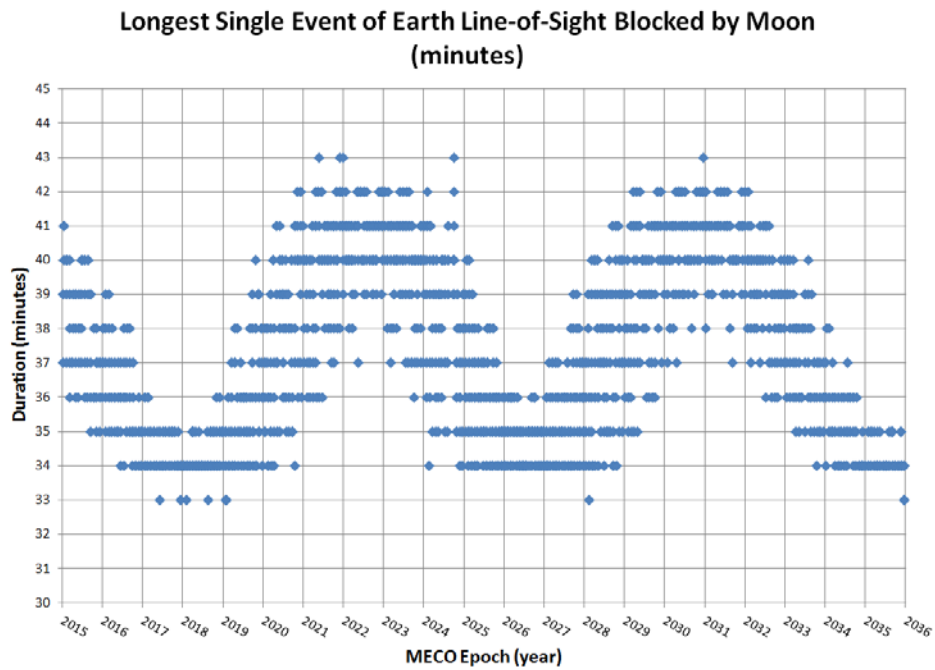


Figure 182. EM-2 Lunar Blockage of Earth Line-of-Sight

Sun-Spacecraft-Earth Angles

Using the assumption that the vehicle is in tail to Sun attitude for a majority of the exo-LEO mission leads to the inevitable importance of the Sun-Spacecraft-Earth angle (see Figure 19) because this shows how close the Earth is (in an angular sense) to the tail of the spacecraft. The tail of the spacecraft is significant because the baseline location of the communication arrays resulted in a gap in communication coverage around the tail of the spacecraft.

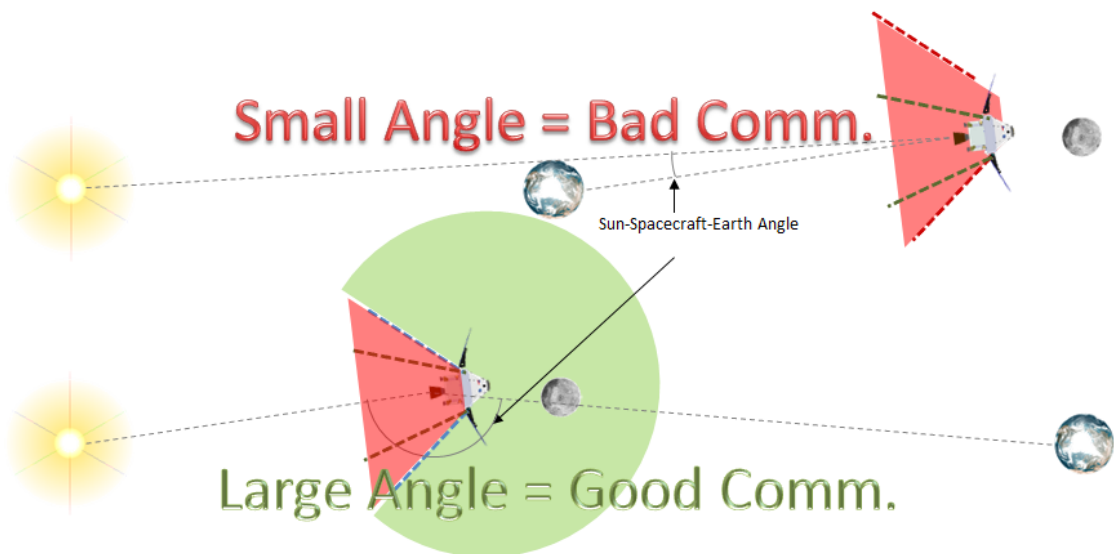


Figure 19. Sun-Spacecraft-Earth Angle

Power

The MPCV power architecture includes power generation/collection performed via four solar arrays located on the SM. The power storage function is performed via batteries located on the CM.

Power margin is calculated based on three primary factors: power usage, power generation/collection, and power storage assumptions. Power usage is driven by vehicle loads, power generation is driven via percent of solar array exposed to sun, and power storage is driven via battery size/capacity required to power the vehicle when limited power generation capability exists.

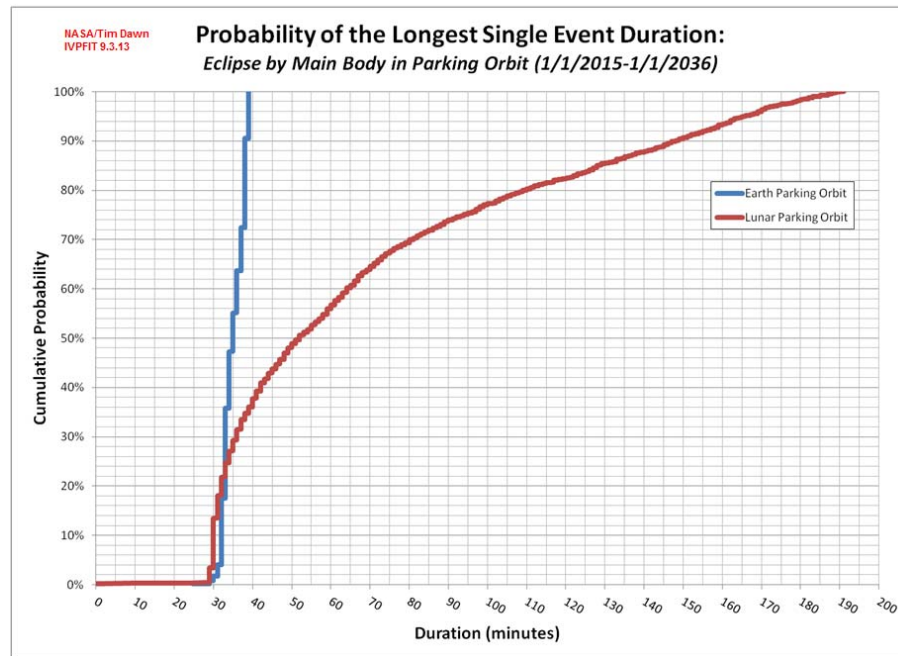


Figure 20. Probability of Longest Single Eclipse duration in Parking Orbit

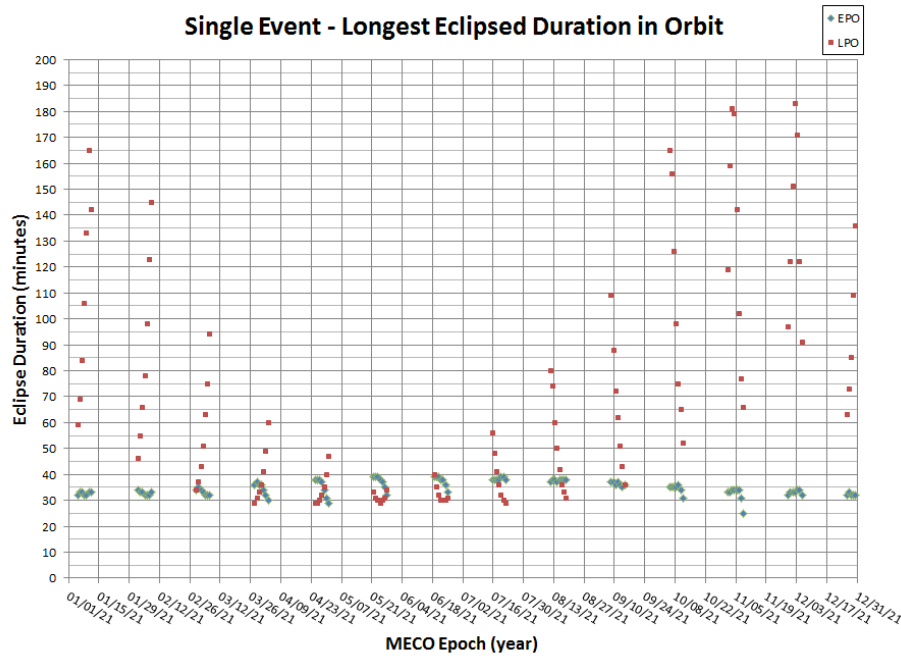


Figure 21. Longest Single Eclipse duration vs. Epoch

Eclipses at Earth and Moon

Lunar and Earth eclipse durations are limited due to existing battery capacity. The spacecraft can only operate without solar array power collection for limited duration before running out of stored energy in the CM batteries. Preliminary sweeps of eclipse durations can be seen in Figure 20 and Figure 2121 . Launch window constraints due to limited allowable eclipse durations are expected.

Eclipses in transit (Outbound/Inbound)

Transit eclipses are bound by similar factors as the Earth/Moon eclipses, where the spacecraft can only operate without solar array power collection for limited duration before running out of stored energy in the CM batteries.

Eclipses prior to TEI

During launch and ascent there is no power collection, so the entire spacecraft is being powered using stored energy in the CM batteries during this time. If on-orbit solar array deployment is shortly followed by eclipse duration, the CM batteries will not be fully recharged. This can translate to low power margin during the high-vehicle-power operations of TEI.

Eclipses near CM/SM separation

After CM/SM separation there is no power collection, so the entire spacecraft is being powered by stored energy in the CM batteries. Longer eclipse durations, prior to CM/SM separation, results in decreased power margin through re-entry and post-landing.

Power

Low Earth Orbit

The external radiative environment of the vehicle is driven by solar heat flux, planetary heat flux (Infrared (IR) & Albedo) and eclipse duration. These are mainly a function of the solar Alpha & Beta angle. For the LEO portions of the EM-2 design space (Ascent & Re-entry), Figure 22 defines the relationship between solar Alpha and Beta angles. The red circles indicate the design points chosen for thermal analysis to envelope the design space. Alpha angles were selected near the 0, 90, 180 & 270 degree points. The corresponding Beta angle extremes were selected for each selected Alpha angle. This captures maximum and minimum eclipse times across the range of solar geometries. For the LEO cases only, the launch epochs associated with each Alpha/Beta combination were used to propagate vector lists for input to the thermal model. Vector lists capture both vehicle trajectories and attitudes.

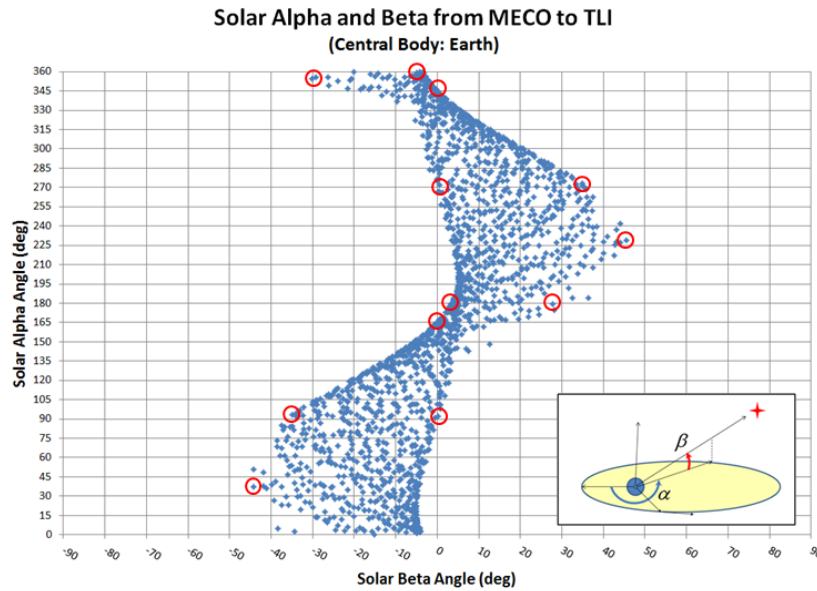


Figure 22. EM-2 HELO Solar Alpha/Beta Space in Earth Orbit

Lunar Orbit

Figure 23 defines the Alpha/Beta relationship for the EM-2 lunar orbit design space. The red circles indicate the design points chosen for thermal analysis to envelope the design space. Similar to LEO, Lunar Alpha angles were selected near the 0, 90, 180 & 270 degree points. The corresponding Beta angle extremes were selected for each selected Alpha angle to capture maximum and minimum eclipse times across the range of solar geometries. The only vehicle attitude modeled was tail-to-sun, which is the baseline.

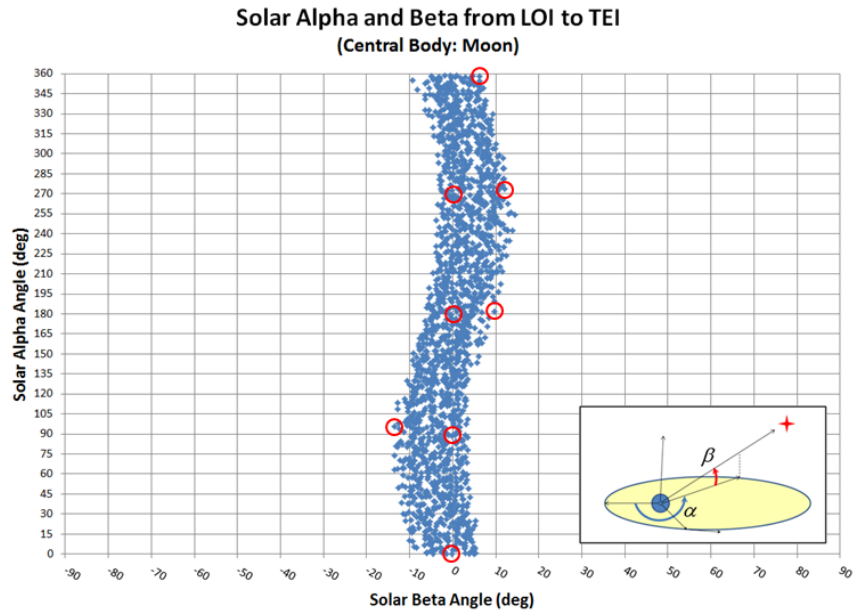


Figure 23. EM-2 HELO Solar Alpha/Beta Space in Lunar Orbit

CONCLUSION

This database of information for the EM-2 mission helps determine how subsystem capabilities and desired operational constraints affect overall mission availability. Guidance, Navigation, and Control (GN&C) systems used this data product to determine which environments it needs to be able to execute in. Prior lunar mission designs involved leaving from a circular Earth Parking Orbit (EPO) and arriving/leaving a circular Lunar Parking Orbit (LPO) during the Constellation Program. This is compared to the elliptical EPO and LPO orbits in this EM-2 mission due to the capabilities of the ICPS and Orion. The elliptical orbits introduced additional geometrical impacts and mission design trends not analyzed before. The analysis over the full lunar nodal cycle allows for a quick comparison of geometrical impacts if this type of lunar orbit mission is performed later in the future. In addition, analyzing the full lunar nodal cycle provides insight into how Earth-Moon geometry affects vehicle performance and mission availability.